SOLAR SAIL VEHICLE SYSTEM DESIGN FOR THE GEOSTORM WARNING MISSION

John L. West†
Jet Propulsion Laboratory
California Institute of Technology
Pasadena, CA

ABSTRACT

This paper describes the mission and system design for a viable, promising first mission application of solar sail technology. The mission – the Geostorm Warning Mission – would utilize small satellite technology merged with a space-inflatable solar sail to take advantage of solar photon pressure to permit a satellite to maintain an unnatural station near the Earth-Sun line at ~0.98 AU. So positioned, such a satellite could offer a factor of 2-3 increase in solar storm warning time compared to a conventional satellite positioned at L1 at 0.993 AU. The mission can be implemented using technology achievable now and readily scaleable to more ambitious future mission applications.

INTRODUCTION

Solar sail technology can enable heretofore nonviable space mission concepts and provide a lower-cost alternative for performing future space missions with high delta-V demands. The technology makes use of the sun's inexhaustible supply of photons to enable missions with non-Keplerian orbits and those that offer unique vantage points. Such missions address a broad range of NASA needs and goals as well as the needs and goals of other federal agencies such as the National Oceanic and Atmospheric Administration (NOAA) and Department of Defense (DoD) with many of these missions emerging from the needs of the Sun-Earth Connection (SEC) theme of the NASA Office of Space Science. Example missions include the Geostorm Warning Mission, the subject of this paper, Solar Polar Imager, and Heliopause Explorer.

The Geostorm Warning Mission is a mission that would provide real-time monitoring of solar activity. It would operate inside the Earth's L1 point and increase the warning time for geomagnetic storms compared to a vantage point closer to the Earth. The concept for the Geostorm Warning Mission originated in the summer of 1996 after NOAA asked the Jet Propulsion Laboratory (JPL) whether an improvement in the warning time available from a satellite positioned at L1 could be achieved through the application of emerging new technologies in solar sails, inflatable structures, and microspacecraft. NOAA's principal motivation was to find a cheap, reliable way to continue the delivery of storm warning data to its commercial and DoD customers after the expected end-of-life of the Advanced Composition Explorer (ACE) spacecraft in 2000-2002. The ACE spacecraft is a NASA scientific spacecraft then scheduled for launch in 1997 which would be positioned at L1 and through agreements with, among others, NOAA, DoD, and NASA, provide – for the first time – continuous storm warning data.

The results of the ensuing JPL study reported at that time in References [1] and [2] showed a viable mission/satellite system concept to provide the desired improvement in storm warning time existed. The satellite could utilize small satellite technology merged with a space-inflatable solar sail to take advantage of solar photon pressure to permit the satellite to maintain an unnatural station near the Earth-Sun line at ~0.98 AU, well inside the L1 point at ~0.993 AU. So positioned, the satellite could provide a factor of 2-3 increase in warning time over the 30 minutes to 1 hour available at L1. The satellite could be based on conventional technology, and the sail could utilize a space-inflated, rigidizable structure.

The mission, as then envisioned, would offer a logical follow-on inflatable structure flight demonstration to the NASA Inflatable Antenna Experiment (IAE) completed in May 1996, taking that demonstration several critical steps further in demonstrating both the deployment of a substantially larger structure than IAE and in-flight structural rigidization. At the same time, the mission could serve an important national need in providing solar storm warning alerts to commercial, DoD, and NASA customers.

The work reported herein summarizes the 1996 work in References [1] and [2] and adds important new detail

† Senior Member of the Technical Staff, Microspacecraft and Advanced Development Group, Mission and Systems Architecture Section

documented in Reference [3] that carries the original 1996 work several important steps further, adding detail to the design of both the sailcraft bus and sail while at the same time validating the original Geostorm system concept and its estimated costs. The latter work in Reference [3] was sponsored by NASA's New Millennium Program (NMP) in the context of a competition for NASA's FY '00 Space Technology 5 (ST-5) technology flight validation opportunity. This work, which led to a formal project proposal which was presented to NASA Headquarters in the summer of 1999, known then as the Sub-L1 Sail Project, was led by JPL and performed with the generous assistance of the Ball Aerospace Corp which developed the details of the sailcraft bus and L'Garde, Inc. which developed the solar sail design.

MISSION DESIGN

Figure 1 shows the key design principles and requirements which guided the mission and vehicle system design for the Geostorm Warning Mission. These requirements address both the needs of NASA to see flight validation of sail technology to enable future NASA missions as well as the needs of two other federal agencies, NOAA and DoD, for acquiring operational space weather data.

- Ensure First Sail Project Success Defined as Meeting the Flight Validation Objectives
- Minimize Risk of Cost Overruns
- Minimize Total Project Cost
- Meet User Instrumentation Measurement Requirements Key of Which Are Magnetic Field Vector Knowledge to 1° and 100 bps Minimum Downlink Data Rate
- Avoid False Alarms and Missed Events
- Provide for Launch After Completion of the Advanced Composition Explorer (ACE) Mission (2000-2002)
- Provide for Achievement of Mission Goals in Presence of Sail Failure
- Serve as a Proof-of-Concept for Subsequent, Additional, Operational Storm Warning Missions
- Provide for 18-Month Operational Mission Life (3 Year Goal)
- Provide Storm Warning Time Better than the ~30 Min Available from a Satellite Positioned at L1

Note: Items shown in priority order

Figure 1. Key Mission Design Principles and Requirements

Figure 2 provides an overview of the baseline mission design for the mission which is described in more detail later in the paper. The figure shows the seven mission phases which would constitute the nominal mission and two additional phases which would constitute an extended mission. All sail flight validation objectives of

Phase	Duration	Validation Objectives Achieved	Comment
• Launch	Hours to days	• None	• Shuttle
• Deployment from Shuttle	Minutes	• None	• Spring ejection from modified Spacelab pallet
• Earth-to-L1 transfer	3 months	• None	• Ballistic transfer using Star 37XFP solid kick motor • Spin mode • Spacecraft propulsion/attitude control S/S provides spin up for and attitude control during kick motor burn and post-burn spin down to nominal 0.3-0.45 deg/s spin rate
• L1 capture	Days	• None	• Non-propulsive
• Sail deployment and characterization	4-6 weeks	• Sail deployment • Sail vehicle sys functionality • Sail effects characterization on spacecraft instruments and systems	• Spin mode
• Conventional L1 mission	18 months to 3 years	• Sail jettison	• Option. Performed only if sail deployment fails
• L1 to Sub-L1 operational station transfer	6 months	• Sail performance as a propulsion device	• Spin mode
• On-station operations	2-3 months	• Sail functionality for in-space stationkeeping	• Spin mode
Total Baseline Mission Duration	12 months	• All	
• Extended mission	18 months to 3 years	• None	• Spin mode. Provides NOAA/AF operational space weather data
• Sail jettison	Minutes	• Sail jettison	• Option after extended mission completion
Total Baseline Plus Extended Mission Duration	2-1/2 to 4 years		

Figure 2. Mission Design Overview

- Sail Deployment
- Sail Vehicle System Functionality
- Sail Performance as a Propulsion Device
- Sail Jettison (Option)
- Sail Effects Characterization on Spacecraft Instruments and Systems
- Sail Functionality for in-Space Stationkeeping

Figure 3. Sail Flight Validation Objectives

interest to NASA, as shown in Figure 3, would be achieved within 12 months after launch during the seven nominal mission phases. Delivery of operational space weather data for NOAA and the Air Force would be accomplished during the extended mission phase with an expected duration of 18 months to 3 years dependent on sailcraft life. Early in this phase, primary responsibility for sailcraft operation would be turned over from NASA to NOAA and the Air Force as operational users of the sailcraft.

Launch

A Shuttle launch is the nominal baseline with a presumed launch date of November '03 per the Reference [3] proposal. Upon sailcraft delivery to low-earth orbit (LEO) by the Shuttle, a Star 37XFP kick stage will kick the sailcraft out to the L1 point.

Orbit

Figure 4 shows the nominal trajectory plot for the mission. The red portion of the path (LEO to L1) is ballistic and takes three months. The green portion (L1 to sub-L1) is a sail trajectory and takes 192 days. The sail orientation (cone angle) is 15° to 50° during the L1 to sub-L1 transit and stays within 5° to 10° once on station at the sub-L1 operational location.

Figure 5 shows sailcraft performance measured by the distance to the Sun at which the sailcraft can maintain an Earth-Sun line station as a function of the sail loading σ, where the sail loading is defined as total sailcraft mass in grams divided by the total sail area in m^2. The figure shows that the sailcraft described herein, utilizing a "conventional" 8-micron thick Kapton sail with a reflectivity of 0.9 and achieving beginning-of-life (BOL) and end-of-life (EOL) sail loadings of 42.1 and 36.3 grams/m^2, respectively, can achieve an operational station location between 0.983 and 0.984 AU. The figure also shows the improvement in station location that could be achieved by the use of a "higher-tech" advanced membrane such as SRS Technologies' ripstop sail which was not selected for the proposal. It is noted, as derived in detail in Reference [4], that for a reflectivity of 0.9 a sail loading of ~28 g/m^2 would be required to maintain station at 0.98 AU, a sail loading not achieved by the design described herein.

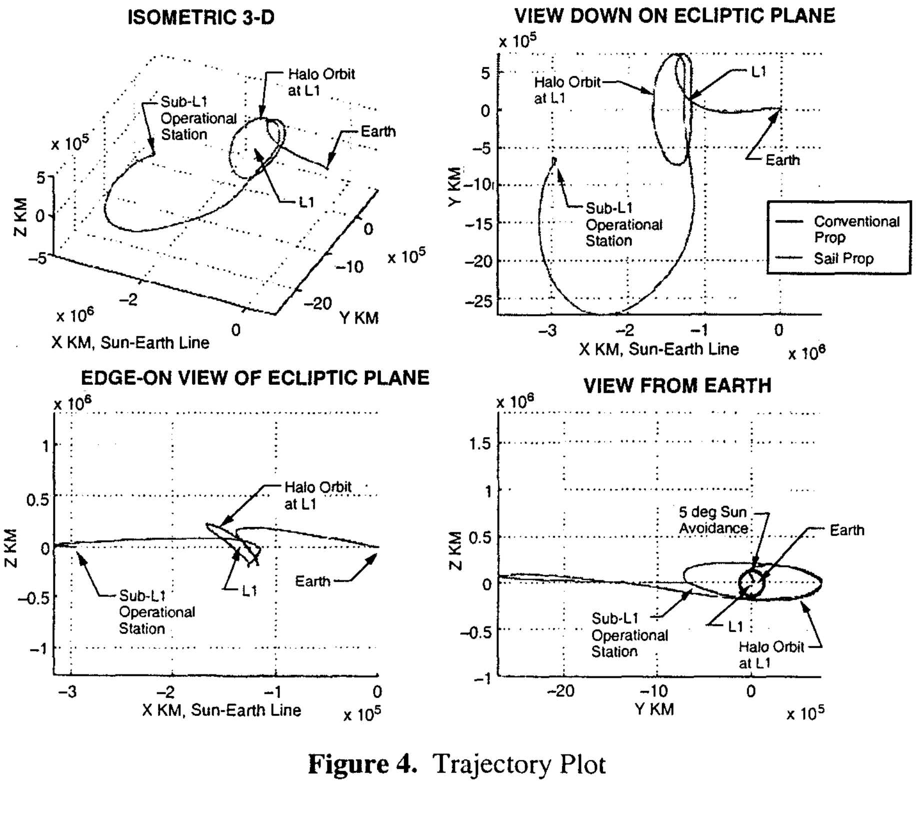


Figure 4. Trajectory Plot

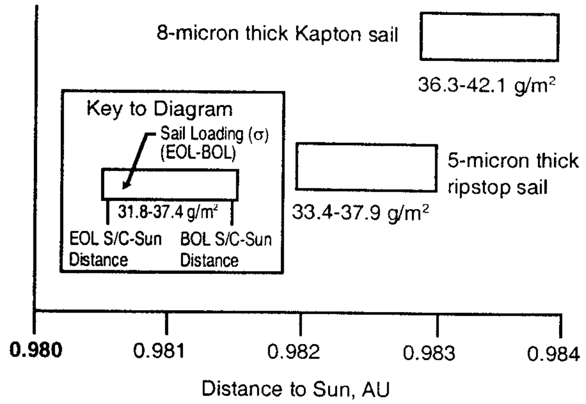


Figure 5. Sailcraft Performance

ΔV Requirements

The ΔV requirements for the mission starting from the circular LEO are shown in Table 1.

Table 1. ΔV Requirements

Maneuver	ΔV (m/s)	Comments
200 km LEO to L1	3200	Star 37XFP responsibility
L1 station-keeping	20	~5 m/s per year required without sail perturbations. Includes margin to accommodate sail
Misc allowances	150	Star 37XFP injection error and trajectory correction maneuvers
Totals	3370	

LAUNCH ACCOMMODATION

Figure 6 shows the sailcraft with its star 37XFP kick stage integrated with the Space Shuttle, the nominal baseline launch vehicle.

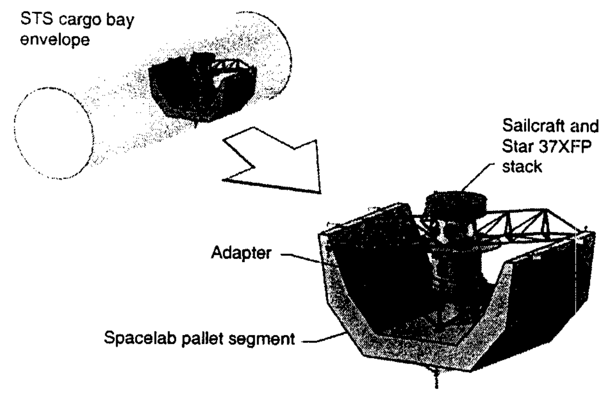


Figure 6. Sailcraft Integrated with Space Shuttle

VEHICLE SYSTEM DESIGN

Figure 7 shows the sailcraft with its sail deployed as it would appear on station at its sub-L1 operational station. The sailcraft is comprised of a three-element instrument payload, the sailcraft bus, solar sail, and sail stowage canister as described in detail in the paragraphs that follow.

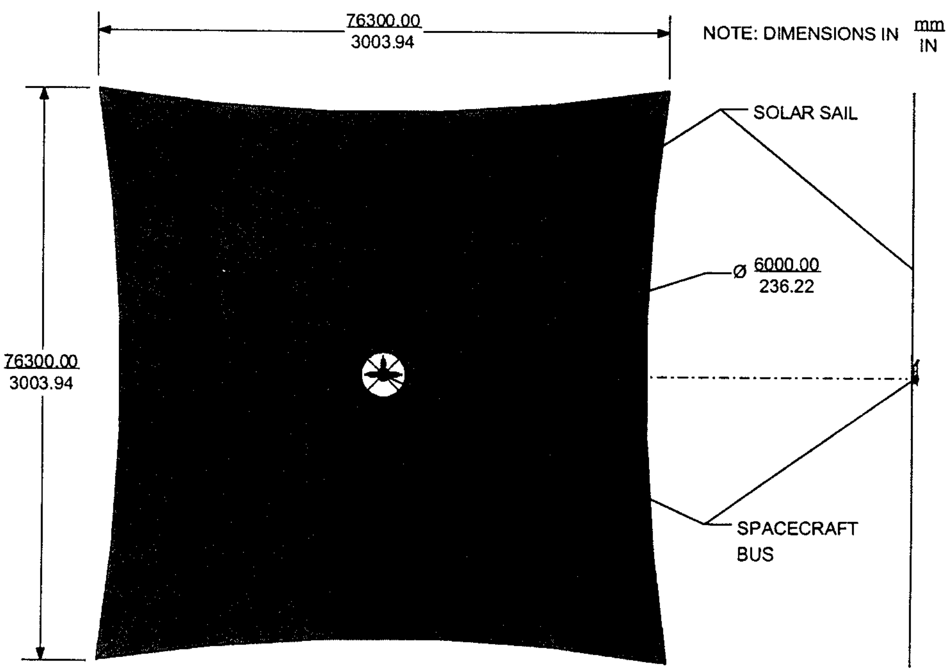


Figure 7. Sailcraft Operational Configuration

The sailcraft utilizes conventional monopropellant hydrazine propulsion to control sailcraft orientation, has a jettisonable sail, and employs spin stabilization for attitude control. Conventional propulsion for sailcraft attitude orientation control was selected to minimize the risk to sail development that would be imposed by the use of other alternatives to orientation control like vanes. Also, the use of conventional propulsion offered, together with the capability for sail jettison, the advantage of permitting the sailcraft to operate – and hence perform a conventional L1 Geostorm mission – without the sail were the sail not to deploy properly and require jettison. In addition to the reason just noted, sail jettison capability was also considered critical to develop and demonstrate to lay the foundation for other sail missions expected to employ the sail as a propulsion stage to be expended upon arrival at a target of interest permitting, for example, high-precision pointing that could be compromised by having a large, difficult-to-maneuver, permanently-attached sail in tow. Finally, spin stabilization was selected after studies of other options such as moving mass systems that provide spacecraft center-of-mass/center-of-pressure control showed these approaches to be more complex and costly than spin stabilization, as discussed in detail in Reference [5].

PAYLOAD

The sailcraft payload consists of operational and diagnostic instruments. The operational package consists of two 3-axis fluxgate magnetometers and an ion plasma instrument. The magnetometers are mounted on the spacecraft thruster booms and have a total mass of 2 kg and a total power consumption of 2 W. They will measure the magnetic field vector to within the requirement of 1 degree. The ion plasma instrument will also have a mass of about 2 kg and a power draw of up to 2 W. It will measure solar wind velocities over at least a 200 km/s to 2000 km/s range, ion densities up to 200/cm^3, and temperatures of 10^4 to 10^7 degrees K.

The diagnostic package will include a camera, several thermal sensors (to be placed on the struts), two load cells on the interior sail catenaries, and a pressure monitor (for use during strut deployment).

If sufficient funds were available, a low-energy electron plasma instrument could be included to measure bow and wake phenomena associated with the sail motion through the plasma.

Degradation of the sail material may be best measured by the change in vehicle spin rate due to mass loss. An independent measurement of the solar flux is not needed since the range to the Sun will be known.

Instrument data will be sent to Command and Data Handling subsystem channels over serial RS 422 lines.

Direct observation of the solar sail at L1 by the Solar and Heliospheric Observatory (SOHO) in an anomaly situation could be considered.

SAILCRAFT BUS

Figure 8 shows several views of the sailcraft bus without the sail in its operational configuration after jettison of the injection stage (IS), and Figure 9 shows the detailed layout of the bus hardware. Key features shown include:

1) The ring shaped solar array which is fixed, avoiding the need for deployment, and which is sized to provide a 28 % end-of-life (EOL) power margin. The ion plasma instrument is mounted inside the open center of the array with both the array and ion plasma instrument continuously sunpointed during on-station operations.
2) The three thruster booms that extend past the periphery of the array substrate which mount the thrusters used for pointing. These booms position the thruster nozzles 2500 mm (98.4") from the vehicle center of mass (CM) to minimize propellant consumption. The forward- and aft-facing omni antennas and the single aft-facing medium gain antenna are mounted to the center boom.
3) The four inflatable struts that support the sail membrane.
4) The sail stowage canister which is supported above the equipment shelf by a cylindrical thrust tube which also supports the battery.

With the exception of the antennas, the thrusters, and the ion plasma instrument, all spacecraft components mount to a single equipment shelf or to the thrust tube simplifying access and cabling layout. The shelf has area for adding redundant units, as required.

The sailcraft bus functional block diagram is shown in Figure 10, and each of the subsystems comprising the sailcraft bus is described below. The sailcraft bus is deliberately designed without new technology to minimize project cost and risk.

Structure

A three-element all graphite-reinforced plastic (GFRP) primary structure has been chosen for minimum mass.

The first element is the 813 mm (32") dia. equipment shelf which is 38 mm (1.5") thick and is fabricated from GFRP face sheets and aluminum honeycomb core.

The 722 mm (30") dia. sail thrust tube's diameter matches that of the injection stage for efficient load transfer. The thrust tube is also fabricated from GFRP sheet.

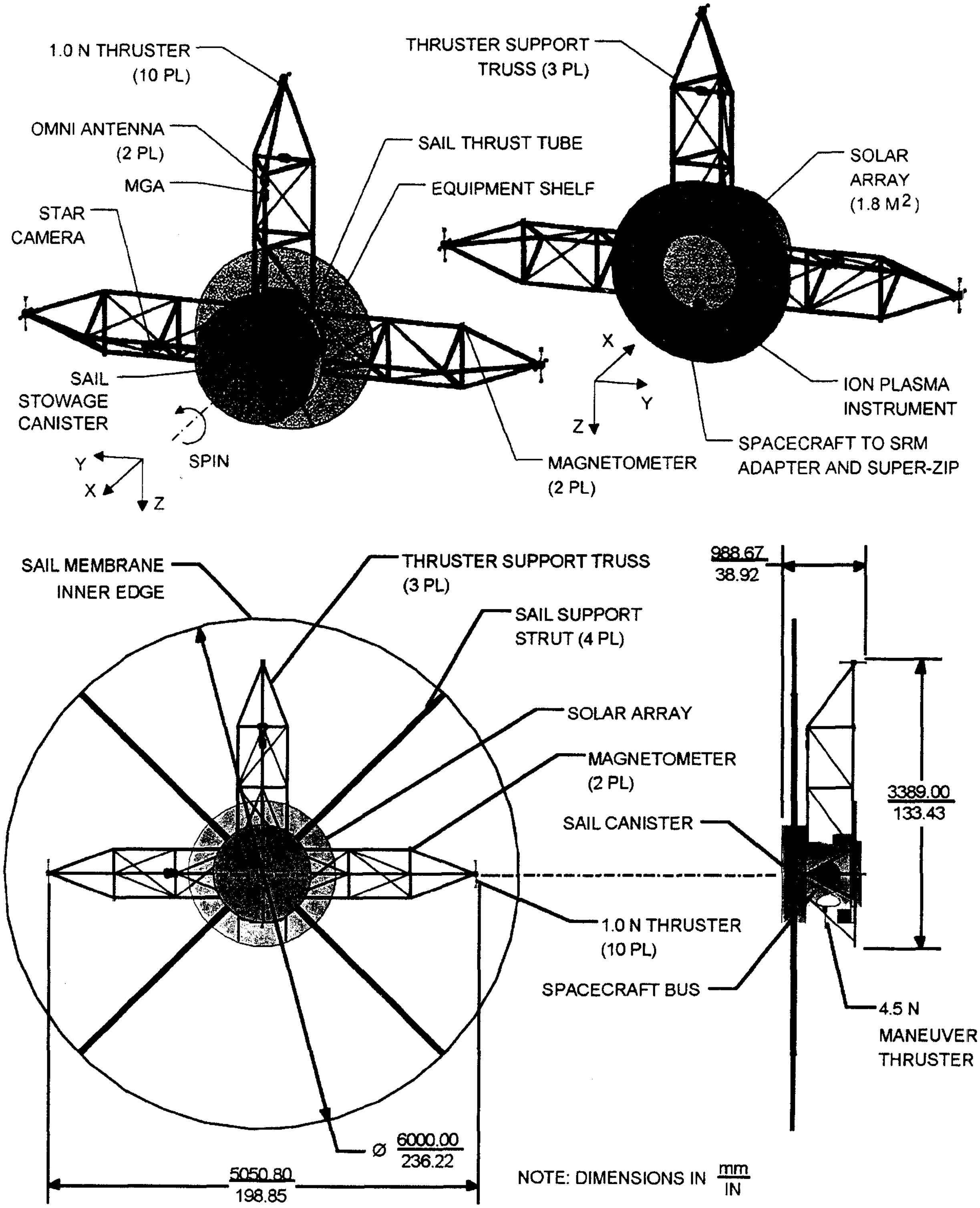


Figure 8. Sailcraft Bus

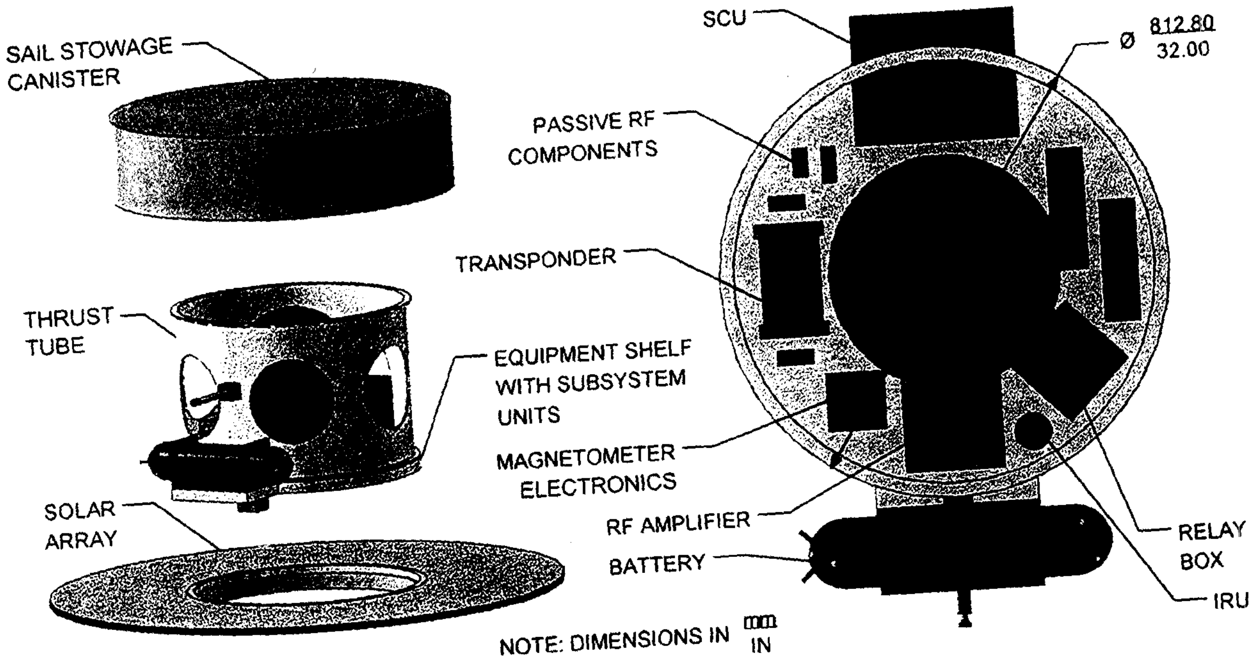


Figure 9. Sailcraft Bus Hardware Layout

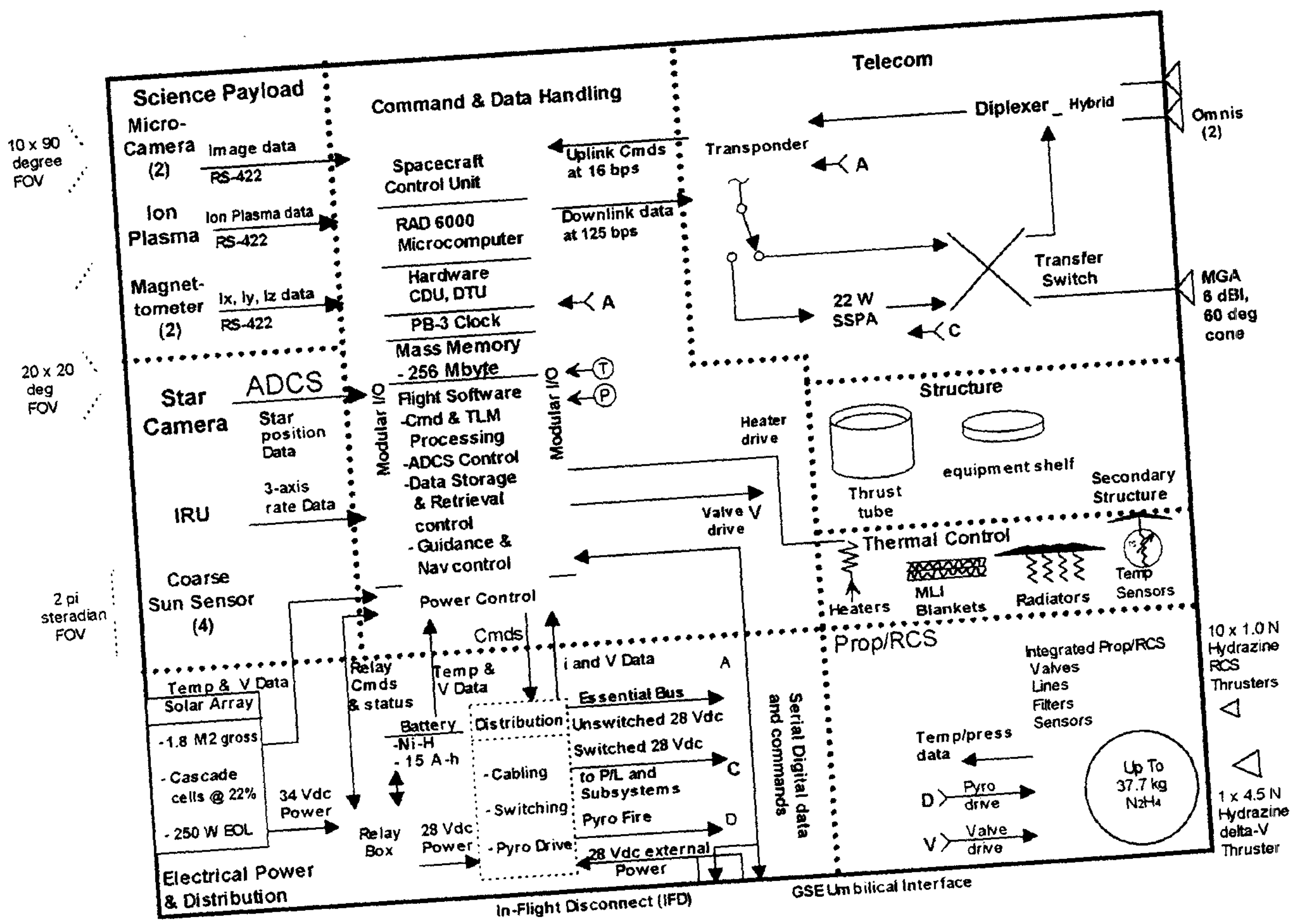


Figure 10. Sailcraft Bus Functional Block Diagram

Three identical 2000 mm (78.7") long truss booms support the thrusters and antennas. Each boom is fabricated from GFRP square and round tubes which are co-cured as a unit for maximum stiffness and minimum mass.

Small secondary structural elements (brackets, etc.) are machined from aluminum bar stock or formed from aluminum alloy sheet.

Power

Power is provided by a direct energy transfer system chosen for low mass. The power system is further simplified by its constant sunpointing orientation and by operation in the mission's shadowless heliocentric orbit.

Command and Data Handling (C&DH)

C&DH is handled by a RAD-6000-based microcomputer which minimizes mass by offloading hardware functions to software. The 8 Mbyte of EDAC RAM on the processor card stores payload data between downloads. The 8 Mbyte provided is sufficient to store more than 20 days worth of data, providing great flexibility in downlink scheduling. Subsystem interfaces are simplified by use of a MIL-STD-1553B data bus.

Telecommunications

Telecommunications utilizes an S-band system to achieve a 125 bps telemetry downlink rate at 0.02 AU range with a 6.7 dB margin using the satellite's medium gain antenna (MGA) and a 22-W output solid state power amplifier. Commands are nominally uplinked at S-band at 16 bps with over 6 dB of margin using the MGA. During a loss-of-attitude anomaly, the two omni antennas would provide for emergency mode downlink telemetry at up to 16 bps as well as emergency uplink. The two omni antennas are also used in geosynchronous transfer orbit (GTO) and during the Earth-to-L1 transfer out to a range of 50,000 km.

Thermal Control

The vehicle's constant sunpointing orientation and operation in the mission's shadowless heliocentric orbit allow mostly passive temperature control which keeps shelf-mounted units between 20 and 40 degrees C using multilayer insulation (MLI) and radiators normal to the sunline. Heaters are provided for the hydrazine lines and tank.

Attitude Determination and Control (ADCS)

ADCS uses a lightweight wide field-of-view star camera and coarse sun sensors to provide attitude determination in three axes to an accuracy of 0.1 deg. Attitude is controlled by spinning the sailcraft about the sail normal axis which points along the sailcraft angular momentum vector. The vehicle's healthy inertia ratio (1:2 between transverse and axial moments of inertia) provides passive spin stability in the presence of perturbing torques induced by offsets between the sailcraft center of pressure (CP) and the sailcraft center of mass. A spin rate of 0.45 deg/s keeps the angular momentum vector (and sail boresight) within 1 degree of the sunline with a 1-meter CP/CM-offset.

To maintain the desired sub-L1 station requires precession of the angular momentum vector at a rate on the order of 7 deg/hr. Precession is implemented by six of the 10 planned thrusters. A healthy sailcraft 2:1 roll-to-transverse inertia ratio allows a particularly simple and fuel-optimal 2-burn precession strategy. The nutation induced by the first of the two precession burns will be canceled by the second burn half a precession cycle (1/4 of the spin cycle) later, effectively eliminating precession-induced nutation. Per Reference [4], propellant consumption for sail control is estimated at around 475 grams/month. Software running in the C&DH microcomputer accomplishes all ADCS computations.

Propulsion/Reaction Control (P/RCS)

The P/RCS needs to provide ~90 m/sec of delta-V for trajectory correction maneuvers (TCMs) on the way to the L1 point. Sail maneuver torque requirements need a minimum thrust level of 1 N from the thrusters. In its RCS role, the P/RCS needs to furnish about 34,000 N-s of impulse for sailcraft spin up and down functions including despin after separation from the injection stage and for sail spin-axis precession and stationkeeping over an ~36 month on-station mission. These requirements are met by a monopropellant hydrazine system operated in a blowdown mode. A single off-the-shelf diaphragm-type tank can hold up to 38 kg of hydrazine. RCS and maneuver thrust impulses are provided by 10 1-N thrusters (Isp = 226 s) and a single 4.5 N delta-V thruster (Isp = 230 s) arranged to provide 3 axes of attitude and 3 axes of translation without requiring reorientation of the vehicle.

Flight Software (FSW)

The flight software takes advantage of the Vx-Works real-time operating system (RTOS) to re-use existing software to reduce cost and risk. Standard FSW functions that are applicable include attitude determination and control, thruster control, command and telemetry processing, and central processing unit (CPU) management. These standard modules are in Vx-Works and are adapted to the mission by updating their databases rather than re-coding.

The estimate of the total number of source lines of code (SLOC) is 35,000 which includes all the modules above. Using a metric of 9.2 16 bit words/SLOC, the FSW load will require about 644 Kbytes of program memory. Available program memory provides > 100 % margin.

The spacecraft processor has a throughput capacity of 35 MIPS. The preliminary throughput estimate for the FSW shows a peak requirement of 8 MIPS, yielding a throughput margin of > 100 %.

Sailcraft/Launch Vehicle Interface

The sailcraft interface to the LV is through the injection stage which in turn mates to a Spacelab pallet and the Shuttle as previously described. The sailcraft mates to the injection stage using Super-zip™ which provides mechanical separation; three separation springs provide a separation velocity of 300 mm/s. Electrical separation is provided by a pair of "rise-off"-type umbilical connectors.

Sailcraft/Science Payload Interfaces

Accommodation of the payload is shown in Figure 8. The two magnetometers are mounted on the thruster booms to minimize EM interference with their operation. The ion plasma analyzer is mounted on the sun side of the spacecraft to provide a close to hemispherical FOV. Both instruments interface to the C&DH subsystem using individual RS-422 asynchronous serial links. Each instrument gets an individual switched and fused 28 Vdc power line.

Sailcraft/Ground System Interfaces

The sailcraft interfaces with the Advanced Composition Explorer (ACE) ground system as described later.

New vs. Existing Hardware

Figure 11 categorizes the required sailcraft bus hardware as flight proven, flight qualified, or new. These categories are defined as shown below.

Flight Proven: This category includes 75 % of the required hardware. It consists of hardware identical to that flown on other spacecraft.

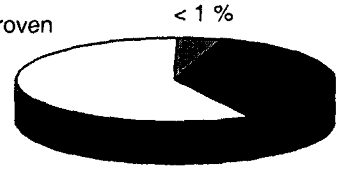


Figure 11. Sailcraft Bus Hardware Heritage

Flight Qualified: This category includes < 1 % of the required hardware. It consists of hardware (or very close derivatives) being developed for other programs with flight dates prior to the planned launch date of the Geostorm Warning Mission.

New Design Needed: This category includes 25 % of the required hardware. It consists of elements normally mission-peculiar or configuration dependent that use strictly conventional materials and design approaches and includes the solar array (assembled from existing, flight proven cells), the GFRP honeycomb structure, wire harness, coax cables, MLI blankets, and RCS subsystem lines and fittings.

Sailcraft Performance and Margin Summary

Mass. Table 2 shows the sailcraft mass by subsystem and includes a blanket 20 % reserve for mass growth. Including reserves, a margin of > 50 % on mass at separation is provided.

Power. Power estimates are summarized for the primary power modes in Table 3.

Performance. Estimated margins provided at the system level are summarized in Table 4. The desired margin values shown in Table 4 are applicable for a Phase A level of design definition, as herein, and are from "Summary of Typical Design Margins and Safety Factors by Project Phase for Unmanned Free-Flyer Scientific S/C" in the NASA Mission Design Process Guide.

SOLAR SAIL

Figure 12 shows the criteria that were applied in selecting and developing the solar sail design described herein. The sail consists of rigidizeable, deployable struts, a thin membrane manufactured from Kapton, and associated inflation/stowage devices and integration hardware. Among these elements, the rigidizeable/ deployable struts are the new technology challenge as is the demonstration of the basic system functionality of a solar-sail-driven spacecraft. Given the significant technical challenges in developing the sail, the sailcraft bus was deliberately designed to include no new technology, as previously noted.

Sail Packaging and Canister Jettison

The stowed sail system is shown in Figure 13. The booms are telescopically packaged with a packaging factor of 2. The stowage canister is a short cylinder on the back end of the spacecraft. Its 1.15-m diameter is determined by the length of the boom packages which are arranged in a star pattern. The sail segments are packaged inbetween the booms. The baseline volume of 0.3 m^3 gives a fairly benign sail packaging factor of 8 for the 0.3 mil thick Kapton sail. Dunnage will be used to assure the membranes will not shift during launch.

Table 2. Mass Summary

Subsystem/Item	Estimated Mass (kg)	Growth Contingency (%)	Mass With Contingency (kg)
Structure	17.0	20	20.4
Power	16.4	20	19.7
C&DH	11.0	20	13.2
Telecommunications	8.78	20	10.5
Thermal Control	2.5	20	3.0
ADCS	1.1	20	1.3
Propulsion/RCS (dry)	11.2	20	13.4
Total, S/C Bus Mass	68.0		81.6
Geostorm Warning Payload (Total)	4.50	20	5.4
Assy, Solar Sail (Total)	78.70	20	94.4
Total, Sailcraft (dry)			181.4
Injection Stage (dry)	119.3	20	143.1
Total, Sailcraft/SRM Stack (dry)			324.5
Propellant and Pressurant			766.0
Total, Sailcraft/SRM Stack (wet)			1090.5
ASE	369.90	20	443.9
Total, Flight System on Spacelab Pallet Segment			1534.4
Spacelab Pallet Segment Load Capability:			3110.0
Margin on Load Capability, kg			1575.6
Margin on Load Capability, %			51 %

Table 3. Power Summary

Item	Cruise/Maneuver Power, W	On-Station Power, W	Launch/Shadow Power, W
Electrical Power and Distribution	4.0	4.0	4.0
C&DH	11.0	11.0	11.0
Telecommunications	9.1	119.1	9.1
Thermal Control	5.0	5.0	5.0
ADCS	16.5	16.5	16.5
Propulsion/RCS	30.0	0.0	30.0
Subtotal, S/C Bus	75.6	155.6	75.6
Electrical Power Contingency @ 20 %	15.1	31.1	15.1
Total, S/C Bus	90.7	186.7	90.7
Science Payload (Total)	6.6	6.6	6.6
Total, Sailcraft	97.3	193.3	97.3
Array Power, EOL @1 Au (Worst-Case)	247.4	247.4	0.0
Margin on S/C Electrical Power, W	150.1	54.1	
Margin on S/C Electrical Power EOL, %	154 %	28 %	
Maximum Shadow Duration, hr			0.6
Shadow Energy Required, W-h			58.4
Energy Required, A-h @ 26 Vdc			2.2
Battery Capacity @ 50 % DOD, A-h			7.5
Margin on Battery Capacity, A-h			5.3
Margin on Battery Capacity, %			234 %

Table 4. Performance Margins

Parameter	Desired Margin	Estimated Margin
S/C Mass at Separation	25-35 %	51 %
Power, EOL	25-35 %	28 %
Pointing Accuracy	x 2	x 2
Knowledge Accuracy	x 1.5	x 20
Propellant Load	30-35 %	30 %
Data Throughput	30-40 %	54 %
Data Storage	40-50 %	> 100 %
D/L RF Link Margin	6 dB	6.7 dB
Torque Factor	x 4	N/A
Strength Margin (Ultimate)	2.1	2.2

The same technique will be used to accommodate transfer stage spinning expected in the 20 RPM range. The canister is vented during ascent.

The canister is held in place by a Marmon clamp. This is released to jettison the canister and enable sail deployment. The clamshell canister walls assure it will clear when released. Jettisoning this mass lowers operational sailcraft areal density.

Sail Deployment

Deployment proceeds in positive and negative directions along one axis and then the orthogonal axis as shown in Figure 14. A blowdown inflation system with a regulated pressure is used for simplicity and lightweight. A latching valve for each axis allows axis sequencing as well as deployment halt in case of an

A. Critical (12 Points Each. Total: 48 Points)	B. Important (4 Points Each. Total: 44 Points)	C. Desirable (2 Points Each. Total: 8 Points)
1. Provides for a Slow, Controlled Deployment Minimizing Film Stress and Film Surface Rubbing During Deployment and, Should Full Deployment Fail to Occur, a Geometry – to the Extent Feasible – Favorable to Degraded Flight System Performance 2. Offers Insensitivity to Flight System Orientation During Sail Deployment, an Unexpected Deployment Sequence, or a Longer-than-expected Time to Deploy 3. Minimizes Total Project Technical Risk, Schedule Risk, and Cost Risk 4. Offers High Structural Margins (Strength And Stiffness) Under Combined Loading and Deflection Conditions, Tailorability to Add Strength Where It Is Needed, and the Potential to Accommodate Less Than Perfectly Straight Struts	1. Minimizes Total Project Cost 2. Accommodates Growth in Sail Dimensions 3. Permits Repeatability in Manufacturing Such that the Results of Ground Analysis and Test Form a Reliable Guide to In-flight Performance 4. Provides Promise for Future Propulsion Subsystem Mass Reductions, that Is, Reduced Areal Mass Densities, as Well as, in Particular, Reduced Structural Element Linear Mass Densities < 45 g/m. Also Includes the Promise for Spacecraft Bus Mass and/or Spacecraft Expendable Reduction 5. Accommodates Spin or 3-axis Flight System Attitude Control 6. Technology Provider Offers Experience and Demonstrated Success with Large, Inflatable Structures 7. Minimizes Membrane Stress Concentrations 8. Minimizes the Potential for Premature Rigidization, Both Pre- and Post-launch 9. Accommodates the Introduction of New Component Technologies 10. Avoids Dependence on Spacecraft Power for Deployment and Rigidization or, if Power Is Required, Minimizes that Dependence 11. Maximizes Maintenance of the Desired Deployed Geometry Under Environmentally- and Flight-system-induced Loads	1. Tolerance to Increasingly Hostile Environments, in Particular, Increased Thermal Loads, Radiation, and Spacecraft-induced Contamination, as Well as Insensitivity to Close Proximity to Either Warm or Cold Structures 2. Minimizes Stowage Volume and Accommodates Stowage Shapes 3. Technology Provider Provides Depth and Breath in Applicable Company Resources 4. Offers Long Shelf Life and Insensitivity to Shelf Stowage Conditions
Notes: • Criteria shown in priority order, top to bottom, left to right • Total potential score: 100 points		

Figure 12. Criteria for Sail Design Selection and Development

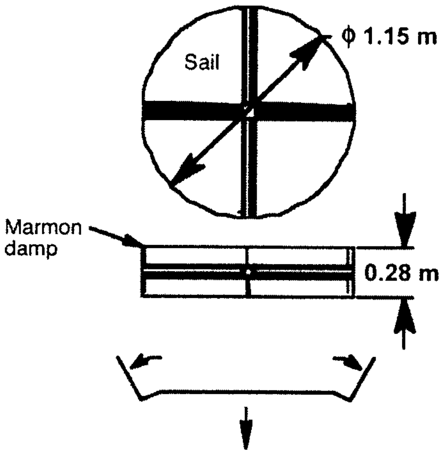


Figure 13. Solar Sail System Packaging and Canister Jettison

anomaly. Contacts at each ring (~1 m intervals) on each boom allow monitoring of all boom/sail positions during deployment. The inflation system is jettisoned after deployment and rigidization to lower sailcraft areal density. The operator will be able to bypass regulated pressure in the unlikely event of a tube hang up.

The sail is attached to the booms via rings at ~1 meter intervals. Therefore, the boom deployment control also controls sail deployment; no mechanisms are added. The Z-folded sail is not pulled across itself or otherwise rubbed. The sail is not crimped other than at the crossfold points. Ample packaging volume exists to pack loosely (packaging factor = 8, as previously noted), obviating these crimps.

If the first axis fails to deploy completely, the second axis of the sail can still be pulled fully out. Also, since the inboard section of sail is deployed first, sail tensioning can still be accomplished in the event of incomplete boom deployment.

Telescopic Packaging and Boom Deployment

The telescopic packaging of the tapered boom is illustrated in Figure 15. This packaging approach minimizes the gas path length for more effective launch venting. Testing has verified that the design packaging factor can be obtained, reference Figure 16. Telescopic packaging and deployment have been flight proven on operational decoys. Ground tests have demonstrated telescopic deployment of a long, shallow cone angle boom as shown in Figure 17.

The longitudinal inflation force, a function of radius 2, is greatest for the outer layer of the telescoping tube. Additionally, the pressure difference to the outside combined with an air bearing effect unblock and lower the friction of the outer layer below that of the inner folds. Thus the strong tendency, verified operationally, is for the outer layer, the base of the tube, to deploy first. This continues as the tube is deployed. This is desirable because the UV window, necessary for rigidization, is deployed first. It is also better to have the inboard sail section deploy first. Further, pressure stabilization of the deployed base sections resists any unforeseen bending loads to keep the boom straight during deployment.

An inner leak-proof bladder allows pressure stabilization for extended periods should interruptions occur. Reserve gas is also carried. Boom deployment can be halted at any time by closing the latching valve. The bypass valve can be used to introduce high pressure if needed. The inner bladder and external insulation also serve an anti-blocking function, keeping the composite layers separated.

Boom Segment Control

For added control, the upper segments are held together until each's turn by a slipstitch, as shown in Figure 18, similar to the type commonly used on bag closures. This technique has been demonstrated in the deployment of a lightweight truss. Pins attached to the

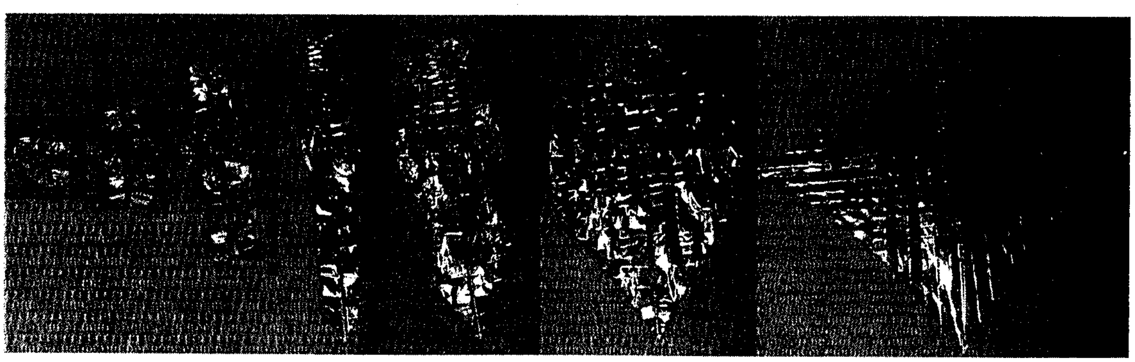


Figure 14. Sail Sequential-Axis Deployment

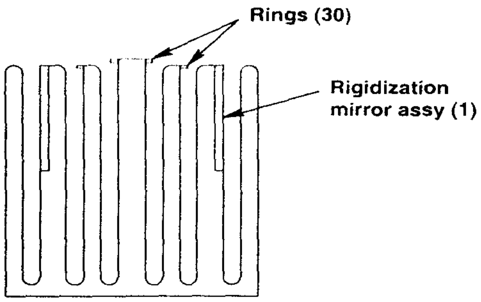


Figure 15. Telescopic Boom Packaging – 50 cm L x 8 cm dia

Figure 16. Packaging Factor = 2

shrouds running along the boom are pulled when a segment nears full extension releasing the next segment from the stack. It also breaks a contact in a bank of parallel resistors of a two-wire circuit running the length of the boom for deployment status monitoring. Weak links and a bypass valve allow override by higher pressure. The tubes have very high burst capability to handle higher pressure. Friction vs. pressure regulates deployment speed inbetween release events. Both sides of the ring will be tied and both must be released before the segment can deploy so the segment will deploy straight. No intelligent controller is relied on nor is power required for deployment.

__UV-Rigidizable Boom Composite__

UV rigidization has been well developed both by L'Garde under the Inflatable Reflector Development and DARPA INSTEP programs and in Europe in the early-to-mid 80's by Contraves using Ciba-Geigy UV-rigidizable resins for their 15-m space rigidizable antenna design.

In the process, initiators in the resin absorb UV light and release radicals which start the exothermic reaction comprising the rigidization process. Heat cannot and

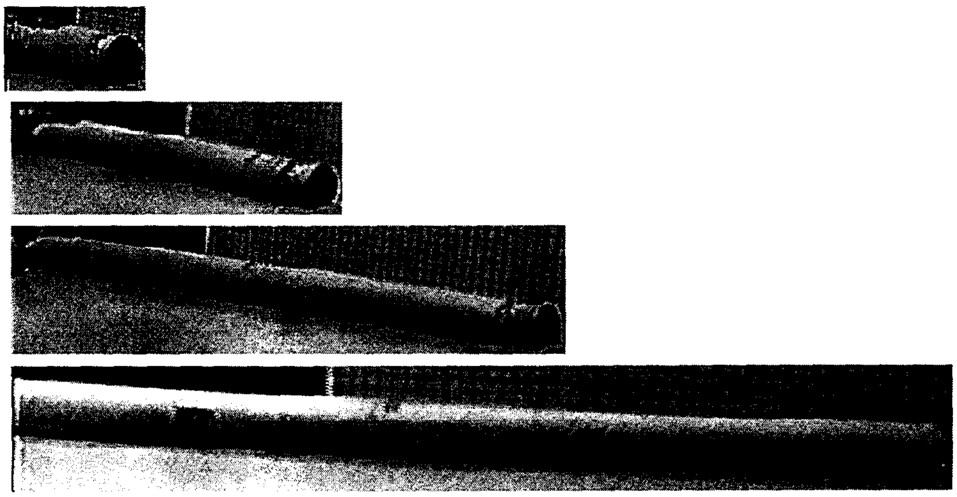


Figure 17. DARPA Long (8 m) Shallow Cone Angle Telescopic Boom Deployment

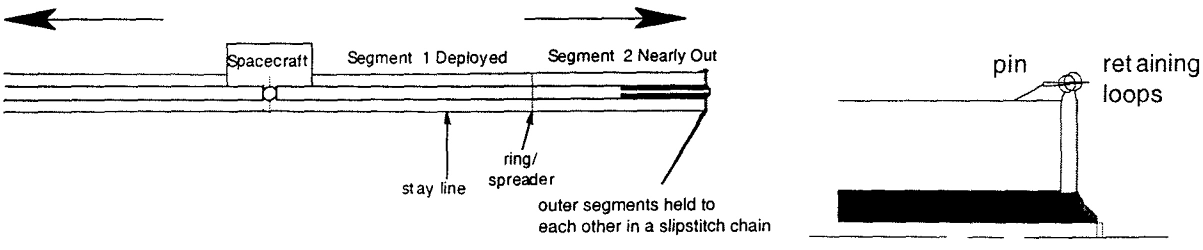


Figure 18. Slipstitch Segment Control

does not either initiate nor affect this process with the material having been rigidized with UV even when it was in a frozen state. This reaction will propagate through 0.04" of material thickness but not 1/8" sideways. If a tear occurs in the outer enclosure during deployment, the UV rigidization will not propagate. The affected area would already be in its desired deployed shape anyway. It would also not propagate through to inner layers, as the insulation is folded in with the boom. Pre-deployment, the canister provides an additional layer of protection.

The UV material has been shown by test to rigidize in low intensity. A low power level can be used over a long time, so long as the total energy is put in. The process is catalytic. Interruptions, as with eclipses, can also be tolerated.

The exothermic nature of the UV rigidization process offers a very helpful by-product: it permits the monitoring and verification of the process. At process initiation the material temperature rises and at completion it falls back to normal. Five temperature sensors will be placed along each strut to monitor and verify its rigidization status.

The UV rigidizable material has excellent shelf life and insensitivity to shelf conditions, as tested. It has also been tested for resin migration in 1 g over extended periods and was found to experience none. It has no latency problem.

Rigidization Control

Rigidization is completely controllable and uses no spacecraft power. Tubes are rigidized from the inside out by the sun via a mirror assembly near the base of each boom as shown in Figure 19. The mirrors initially block the aperture. Once the booms are deployed, while still inflated, both redundant mirror sets are commanded open via latching paraffin actuators. The mirrors reflect light 90° down the tube to rigidize the boom walls through the bladder. The UV material has been tested to verify that it will rigidize through the bladder. The amount of sun energy required is also low (15 watt hours per 50-m boom), and low flux levels are acceptable, so the mirror aperture is kept quite small. Time required is 36 hours. More initiators can be added to the UV matrix to speed the reaction.

The mirror set is deployed first with the boom and is located 1.25 m away from the spacecraft center to avoid shadowing. Its thickness is accommodated in packaging by increasing the boom diameter below the mirror set.

The fact that the tube is tapered is advantageous. If it is perfectly straight as desired, light simply illuminates the

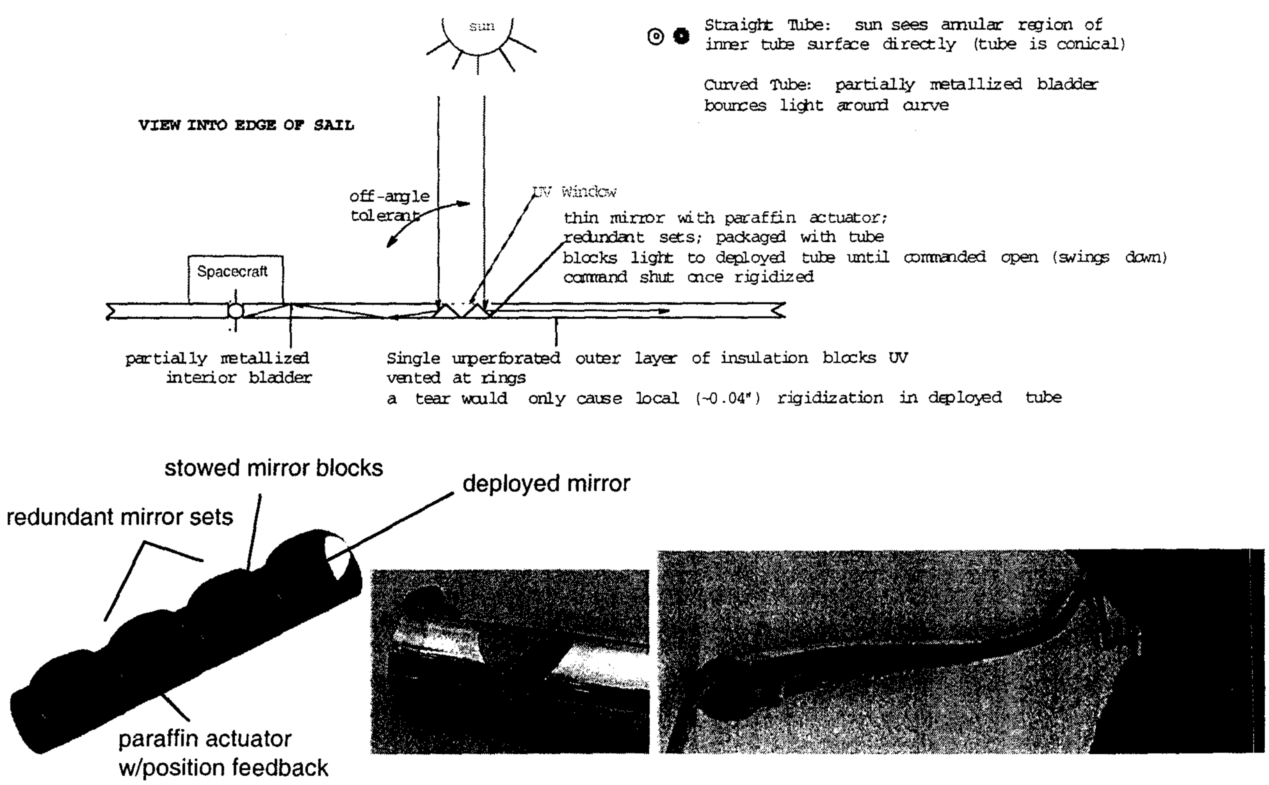


Figure 19. Light Distribution Optics Design and Demonstrator Optics

annular region of material it sees from the base. However, in case there is any boom curvature, the bladder is partially metallized to reflect some light down the tube and around any curves. The degree of metallization is varied along the boom to get an even illumination. A mirror at the tip prevents leakage. This light distribution technique is used in commercial homebuilding to redirect skylights. A full scale 50-m bladder light guide was tested by L'Garde demonstrating excellent light distribution to and transmittance through the tip even with the tube heavily bent. A thin-film mirror was used. Sun off-pointing angles up to ± 15° were also tested successfully. In the event a tube does not deploy fully, the exposed insides would still rigidize for structural integrity. Redundant mirror sets with separate command and power lines are used to guard against the possibility that one set will fail to deploy although even a partially deployed mirror will still distribute light. Once rigidization is complete, the mirrors are commanded closed, and the inflation gas is vented through null jets. If there is a failure to close the mirrors, the small added heat in the boom will raise its temperature, but not by much, especially since only one layer of insulation is used. This would not cause a problem for the boom: the material was tested at high temperatures with no strength degradation.

Boom Thermal Gradient Control

During and after deployment, the deployed boom segments are protected against thermal warping by a layer of insulation. The temperature difference across the boom diameter is reduced from a delta of 65° C to a delta of 4° C. This layer also keeps the undeployed material flexible during any eclipse periods. Additionally, it keeps light out, preventing rigidization by the sun. As such, it is not perforated, but is vented through passages in each of the 30 rings.

Boom Compressive Load Capability and Insensitivity to Length and Curvature

The minimum sail stress of 1 psi (area averaged) in a 0.3 mil Kapton sail ~70 m on a side produces a compressive force in the boom of 2.3 lbs for a simply supported sail or 3 lbs for the catenary design herein. A thinner sail material would give proportionately lower load with appropriately higher safety factors. The sail spin rate of 0.45°/sec is for attitude stabilization; it only offloads 0.008 lb. The structural design will work spinning or not with large margin.

The 4.5 mil UV/Kevlar composite, a seamless tube, will resist 30 lbs in short cylinder compression, giving a large margin (safety factor = 10). This capability is a function of wall thickness, not length or radius. The tube is 8 cm in diameter at the base tapering to 2.5 cm at the tip. It weighs 41 g/m including the composite, bladder, insulation layer, ring/stays, and shrouds. An option is to use a 3 mil composite, giving a linear mass of 32 g/m with a safety factor of 3.8. The composite thickness can be scaled higher for added strength. Modulus would also improve with development, but the current modulus of 1.2 Mpsi is adequate. The large margin allows operation well below 3 sigma, reducing risk.

The problem of long column buckling is eliminated by attaching the sail to the boom along its length using rings, giving a follower loading condition. This is necessary because the need for low areal density forces a small diameter, long boom. The resulting high slenderness ratio (L/R > 1000) is at least an order of magnitude above conventional column structures. The long column (Euler) buckling of an untapered column is a strong function of both length and radius, that is:

$$P_{euler} = n^2 \pi^2 EI / L^2$$

for a column with hinged ends

where n = mode
(= 1 if sail is only attached at the tips)

$I = \pi t r^3$

E = modulus

L = length

This assumes low end fixity typical for such slender beams. If the sail is attached only at the base and tip, the mode number n=1. By attaching the sail at ~1 m intervals along the boom via rings, n is increased to 30, which greatly increases capability (30^2 = 900X). The load is allowed to follow the boom as it bends, a "follower" load. In fact, if the sail were attached continuously along the boom, there would be no Euler buckling instability. This is the case with the sail downhaul force on windsurfing masts. The rings are also used to stabilize the tube cross section shape, as well as for deployment control and out-of-plane stiffening.

The boom is tapered so the rings need to be more closely spaced near the tip. A Finite Element Analysis (FEA) was run to determine the required spacings and number of rings (30) to achieve a 50 lbf Euler capability as shown in Figure 20. This greatly exceeds the short cylinder capability, so the boom is no longer limited by Euler buckling.

Note from the equation that P_{euler} is a function of the squares of both n and L so this load does not decrease for longer booms, as long as the attachment spacing is

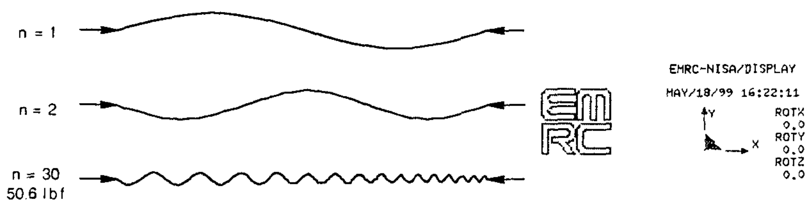


Figure 20. Finite Element Analysis Results for Tapered Boom

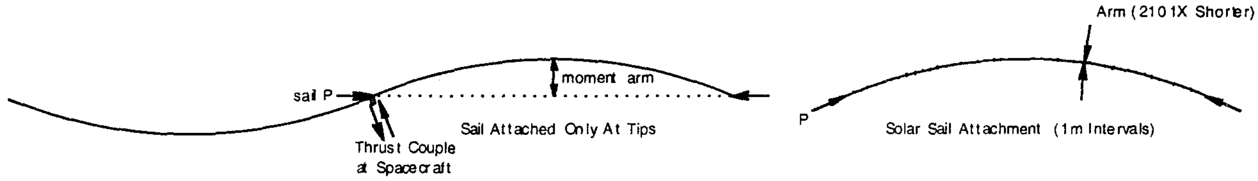


Figure 21. Relative Moment Arms Due to Curvature for Different Sail Attach Method

kept the same (rings are used on the boom extension). This means that the design is length insensitive. The applied sail force does increase linearly with sail size, of course, but this can be handled with the large short cylinder buckling margin. This effect also works during deployment and prevents "jack knifing." The sail is attached to the boom periodically, so any unforeseen sail force is still a follower load.

Another advantage of follower loading is insensitivity to boom curvature. Curvature results from manufacturing, deflection under bending loads, and thermal warping. Curvature presents much greater moment arms to booms with sails attached only at the ends than to follower setups, as illustrated in Figure 21. Straightness due to manufacturing and deployment is usually modeled as a "transition curve" between short cylinder buckling and the Euler limit curve. Straightness becomes worse for higher L/R. Some curvature is present for all slender booms, even precision machined booms. Thus, the insensitivity to curvature offered by periodic attachment is crucial for such a long, slender boom, especially under combined load and deflection conditions.

Isotensoid Sail Suspension

The catenary sail tensioning system places the sail in isotensoid stress as shown in Figure 22. The booms pull on edge cords, not directly on the sail material. The material is loaded evenly as with the deck of a suspension bridge. This means that there are no stress concentrations and no stress wrinkles. The lack of stress concentrations provides margin for fragile, thinner sails and/or higher desired average stress. The catenary edge cords are very light.

A slightly higher load and longer booms are necessary with catenaries vs. simply supported sails with equal area. The deeper the scallops (the longer the booms), the lower the load would be, down to a limit. This design places 3 lbf in each boom compared to 2.3 lbf for a simply supported sail. This can be adjusted but is worth doing considering the wrinkle-free nature of catenaries and the lack of stress concentrations. A thinner sail would use the same geometry but would yield proportionately lower boom forces. An example of catenary hardware is shown in Figure 23. This design was also used for a waveguide.

Although the solar pressure is low, it still requires the sail to billow to a 3D shape in order not to have infinite loads.

This system allows for equal contraction and expansion under thermal variations as well as creep minimization. In eclipse, the solar pressure would be off, so the 3D sail camber can be taken up as slack to compensate somewhat for sail contraction. Also, the flexible sail attachments have sufficient degree of freedom to allow the sail to thermally expand and contract relative to the booms. The sail is attached to the boom along its

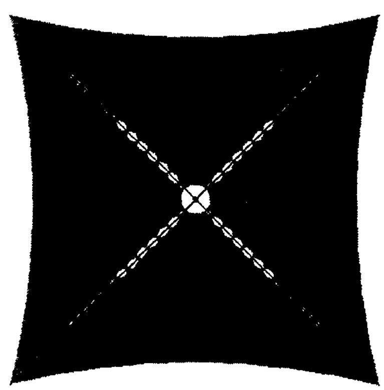


Figure 22. Catenary Sail Tensioning System

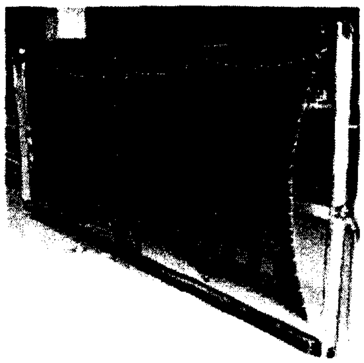


Figure 23. SAR Catenaries
(11 mil flatness; no wrinkles)

length, so any unforeseen sail contraction forces would load the boom in a following manner taking advantage of the ample short cylinder structural margin. Creep will also happen evenly, and the sail ties will take up the slack as the sail creeps over time albeit with somewhat lower stress. This would allow some investigation of the effectiveness of sail stress on solar attachments have sufficient degree of freedom to allow sailing during the mission. Load cells will measure the catenary cord tension so the sail stress can be relayed to earth.

Sail Membrane, Ripstop, and Reflective Coating Properties

The sail membrane is made of 0.3 mil Kapton flat sheets cut into gores, butt-jointed, and taped. The seam tapes are effective as ripstop. Additional (thinner) tapes are added in the cross-wise direction to prevent tear along the length on a gore. The ripstop thus forms a boltwidth wide square pattern. The front side is metallized. L'Garde has measured reflectivities from 0.3 mil Kapton with 300-600a vapor deposited aluminum (VDA) between 0.85 and 0.9. It may be possible to increase this up to 0.95 using coated silver. The backside reflectivity of this Kapton, without any emissive coatings, was measured as 0.58 to 0.7, so a reasonable temperature can be maintained. The sail attachments have sufficient degree of freedom to allow the sail to thermally expand and contract relative to the booms during and after deployment.

Kapton is well flight proven and commercially available with excellent mechanical properties. Currently, the thinnest available is 0.3 mil, but thinner material could be made for future sail missions. A 6-meter hole is used to avoid thruster plume erosion. Larger holes can be accommodated if necessary.

The crosswise ripstop is added only as a measure of prudent design. Once a rip is started, it will propagate easily in this thin material, but it does need a propagation source. The only conceivable sources are handling (for which it would simply be patched), deployment snags, and micrometeroids. However, the packaging and deployment are rather benign, and tests of micrometeroid impact on thin films under tension show no tear propagation.

If a tear does occur, it will be contained within a ripstop square. This slit will have minimal impact on sail area. To further limit the impact, the ripstop is designed with rip terminators to prevent rip propagation along a seam as shown in Figure 24. The packaging and mass impacts of this ripstop system are minimal.

Out-of-Plane Stiffeners

Out-of-plane stiffeners are used on the booms for three main reasons:

- To resist the tendency for the booms to go out of plane
- To increase boom – therefore system – natural frequency
- To increase bending safety margin

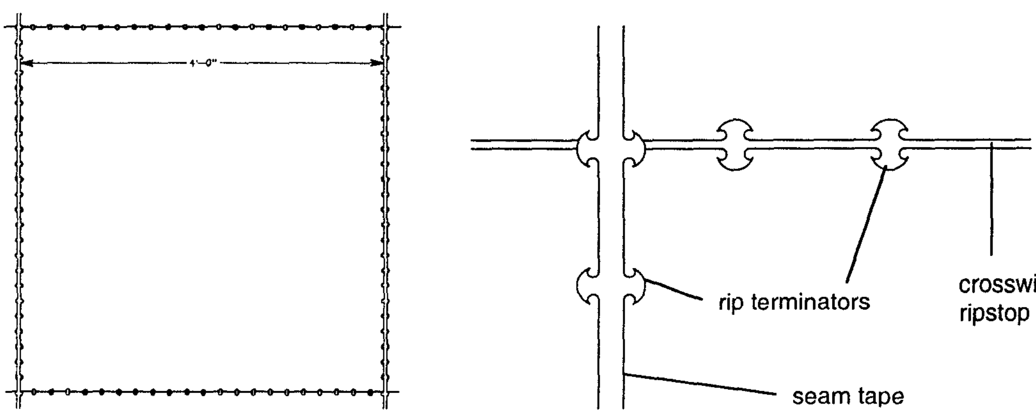
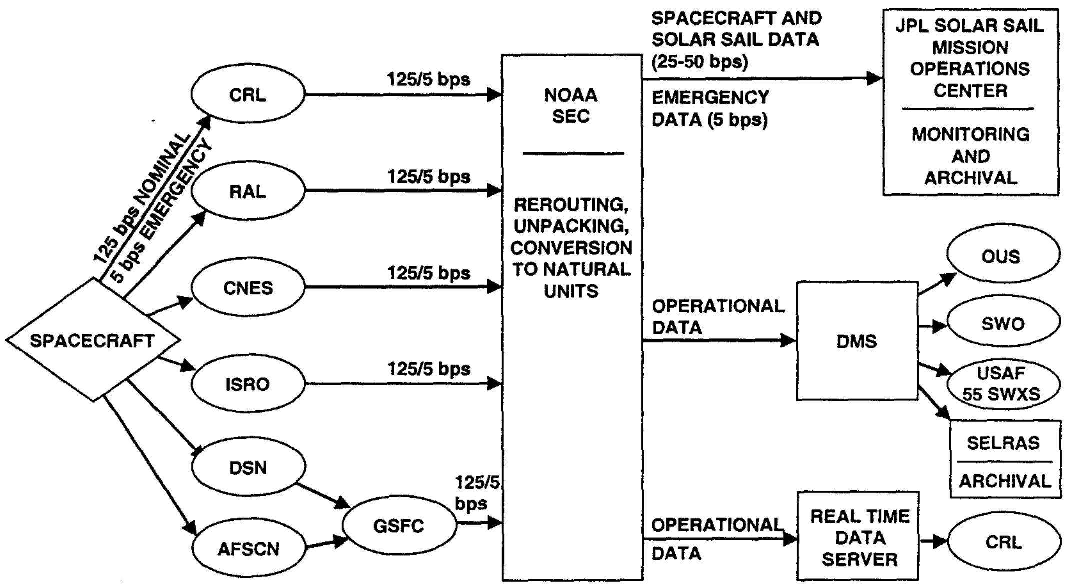


Figure 24. Rip Terminators Guide a Rip Back onto Itself to Stop It

Table 5. Sail Propulsion Subsystem Mass Summary

Component	4.5 Mil Boom	Optional 3 Mil Boom
8-micron Kapton Sail + Catenaries	57 kg	57 kg
Boom Composite	28.3 g/m	18.9 g/m
Bladder/Light Distribution	1.4 g/m	1.4 g/m
Insulation	2.8 g/m	2.8 g/m
Ring/Spreaders and Shrouds	8.6 g/m	8.6 g/m
SUBTOTAL	41.1 g/m	31.7 g/m
4 BOOMS TOTAL	8.9 kg	6.8 kg
Mirror Assembly (x4)	2 kg	2 kg
Instrumentation	3.5 kg	3.5 kg
Sail Jettison System	0.25 kg	0.25 kg
OPERATIONAL TOTAL:	71.7 kg	69.6 kg
Inflation System (jettisoned)	2 kg	2 kg
Stowage Canister (jettisoned)	5 kg	5 kg
LAUNCH TOTAL:	78.7 kg	76.6 kg



ADVANCED COMPOSITION EXPLORER (ACE) GROUND SYSTEM PLUS JPL-BASED OPERATIONS CENTER

Figure 27. Ground System Architecture

When sufficient data (16 to 64 seconds) have been collected, the processing programs will convert the data into natural units (nanoTesla for the magnetic field strength, cm^{-3} for ion density, K for temperature, and km/s for solar wind speed) and write these values to the SEC Data Management System (DMS) database. The data will also be written to an area of preprocessor memory for use by averaging programs.

The x-, y-, and z- components of the magnetic field will be calculated, along with latitude and longitude, in 16-second intervals, while density, temperature, and solar wind speed data will be calculated in 64-second intervals. Blocks with missing data will be thrown out. The data will be processed and dispersed within five minutes of the time it leaves the sailcraft so that timely warnings can be provided of impending geomagnetic activity. Data will be routed to SEC Space Weather Operations (SWO), the SEC Outside User System (OUS) for general operational user community access, the USAF 55th Space Weather Squadron (55 SWXS), and Tokyo's Communication Research Laboratory/Hiraiso Solar Terrestrial Research Center Regional Warning Center. In addition to this real-time use, all RTSW data will be archived in the SEC internal data store, SELRAS, for long-term operational analysis. SWO will provide qualitative summaries of the solar wind information in its daily *Report of Solar and Geophysical Activity,* the *Solar Coronal Disturbance Report,* and the weekly *Preliminary Report and Forecast of Solar Geophysical Data.* The data will also be used to continuously generate a predicted geomagnetic activity index, comparable to the geomagnetic Kp index, that will be used as guidance for issuing a warning of an expected disturbance.

The software for all these tasks already exists and is in use.

MISSION OPERATIONS

The mission is divided into several phases: launch, transit to L1 capture, sail deployment, transit to duty station/initial stationkeeping, operational stationkeeping, and possibly end-of-life testing.

Commanding of the sailcraft will be performed from JPL for the first few phases with control nominally turned over to NOAA and the Air Force following completion of the validation objectives during the 12-month nominal mission.

The launch phase will include the Shuttle launch and the spring ejection deployment of the spacecraft from a modified Spacelab pallet. It will be performed in a non-spin mode.

The transit to L1 will be ballistic, using a solid kick motor. The spacecraft will be spun up for the kick motor burn and spun down to roughly a 0.45 degree/second spin rate after the burn. During the transit, the real time solar wind sensors will be turned on and checked out. This phase will last for three months. After the first month, partial use of the instrument data for NOAA forecasts will begin. JPL will monitor the spacecraft only during prime shift at this time (with 24-hour monitoring available during anomalies).

The sail deployment phase will take place at L1 and last 4 to 6 weeks. It includes strut rigidization. Initial sail maneuvering will begin, as will full forecasts using the real time solar wind data. JPL will again monitor the spacecraft only during prime shift, with the exception of 24-hour monitoring for the day of sail deployment, the day of rigidization, and the day of an initial sail maneuver. Once again, the spacecraft will be spun at roughly 0.45 degrees per second. Since the real time solar wind instruments will already be functioning, this phase will serve to validate not only sail deployment and sail-vehicle functionality but also the effect of the sail on the solar wind instruments. If the sail deployment fails, the sail may be jettisoned so that a conventional L1 Geostorm Warning Mission can still be performed.

The transit to the duty station will take approximately six months. This will validate the performance of the sail as a propulsion device. The spacecraft will remain at the same spin rate and will be monitored during prime shift. This phase will include a few weeks of initial stationkeeping. The next phase will be operational stationkeeping. During this time, command and control will nominally be transferred to NOAA or Air Force personnel or their contractors. Should the spacecraft remain healthy until a second Geostorm spacecraft is made operational, there could be end-of-life testing, including a sail jettison.

Uplink will be S-band at 16 bps. Instrument commanding will be done on an "as-required" basis.

During nominal operations, staffing will consist of three full time personnel: a Mission Planner, a real-time Operations Engineer, and a Downlink Engineer. Varying amounts of part-time support will be needed from a Navigator, Attitude Control Engineer, and other subsystem engineers. JPL will not provide a full-time Instrument Engineer, as payload questions will be dealt with by NOAA.

The Mission Planner will be responsible for overall scheduling of mission events, including ΔV maneuvers, attitude changes, calibrations, imaging, and operational changes due to increased range. The planner will produce up-to-date short term and long term schedules and mission plans. The Real Time Operations Engineer will generate, validate, and maintain sequence blocks, stored sequences, contingency sequences, and command loads and uplink them to the spacecraft. The Downlink Engineer will monitor spacecraft and sail health, analyze diagnostic data, and lead anomaly investigations and recovery teams.

There will be a set of subsystem engineers available to join any required anomaly team. This includes specialists in attitude control, telecom, command and data handling, propulsion (hydrazine), power, thermal control, navigation, contamination, space environmental effects, mechanical systems, solar sail subsystem, and instrumentation.

Table 6. Risk Management Approach

Risk Category	Key Risks	Mitigation Approach	Comments
• Cost	• Overruns	• Relax required characteristic acceleration: – < 0.28-0.30 mm/s^2 • Provide 20 % reserve on all project elements, higher reserve on higher risk elements	• Applicable to all risk categories. Mitigates mass growth as a cost, schedule, technology, and engineering threat
• Schedule	• Slips	• Relax required characteristic acceleration • Provide 1 month reserve per year of phase C/D schedule	• Same as cost
• Technology		• Relax required characteristic acceleration • Get alignment of government/commercial R&D efforts to enhance TRLs	• Applies to all technology risks
	• Sail propulsion subsystem functionality	• Perform extensive full-scale model development and testing under expected environmental conditions	• Top three risk
	• Strut functionality	• Same as above	
	• Mirror/bladder system functionality	• Same as above	• Top three risk
• Engineering			
– Flight System	• Any	• Relax required characteristic acceleration • Staff Mission Assurance function early • Plan for peer reviews of critical and/or new designs	
• Hardware	• Margins exceeded	• Maintain mass and power margins > 20 % during Phase A • Maintain 100 % memory and performance margins during Phase A	
	• Component failures	• Design for functional redundancy • Utilize selective "high impact" component redundancy • Avoid deployable devices	
• Software	• Navigation software functionality	• Perform extensive ground simulation, testing, and analysis	• Top three risk
– Ground System	• None major	• N/A	• Backup uplink/downlink capabilities will be identified
– Launch	• None major	• N/A	• Generous mass margins provided for
– Operations	• Sail failure to deploy	• Spacecraft designed to permit sail jettison and performance of a conventional L1 solar storm warning mission	• Has NOAA/Air Force buy-in

RISK MANAGEMENT

The risk management approach is shown in Table 6. This table breaks the risks to mission success into four major categories, identifies the key risks, and shows the mitigation strategies for the risks. It also identifies the top three risks and the approach to their mitigation.

Fundamental to the risk management strategy is the acceptability of relaxing the required characteristic acceleration of the sailcraft. This flexibility, together with generous mass margins vis-a-vis both the Shuttle launch vehicle and the Star 37XFP kick stage, remove sailcraft mass growth as a major threat to cost, schedule, and sail propulsion subsystem development.

Use of this approach to risk mitigation is permissible because of sponsor and user willingness to accept the slightly lower solar storm warning times associated with a lower performing sail in the interest of seeing the technology successfully demonstrated.

SUMMARY AND COST

This paper has described the mission and system design of a viable, potential first mission application of solar sail technology which usefully exploits the performance and cost advantages afforded by solar sail technology to serve an important national need while at the same time providing a convenient test bed for demonstrating solar sailcraft viability in space. It is based on technology achievable now and readily scaleable to more ambitious future mission applications.

The extensive costing efforts reported on in Reference [3] suggest a total mission life-cycle cost in real-year dollars for a Project start in FY '00 of $88 M including 20 % reserves and Shuttle launch vehicle integration costs but excluding the cost of the Shuttle launch itself.

ACKNOWLEDGEMENTS

The research described in this paper was carried out ~~by~~ at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration. It was sponsored by NASA, NOAA, and the United States Air Force.

Special thanks go to Mr. Richard Reinert and the Ball Aerospace Corp for developing the details of the sailcraft bus and Mr. Billy Derbes and L'Garde, Inc. for developing the sail subsystem design under contracts with the Jet Propulsion Laboratory.

Thanks also go to Dr. Chen-wan Yen of JPL for mission design, Dr. Gurkipal Singh of JPL for attitude control analysis and design, Mr. Steven Tyler of JPL for payload, ground system, and mission operations definition, and Mr. Bain Dayman of JPL for launch vehicle integration.

Acknowledgement is also made to Ms. Patricia Mulligan and Dr. Ronald Zwickl of NOAA and Maj Russell Kutzman of the Air Force Space Command for user requirements definition, to Dr. Colin McInnes of the University of Glasgow for initial mission concept viability studies, to Dr. Robert Forward for early technical brainstorming discussions, to Ms. Lesley Rahman and Allied Signal for ground system definition support, to Mr. John Martinec and Teledyne Brown Engineering for launch vehicle integration technical and cost studies, to Mr. Chris Talley and SRS Technologies for providing a "high-tech" sail membrane option for Team consideration, and to Mr. David Cadogan and ILC Dover, Inc. for participating in the sail provider downselect process.

REFERENCES

[1] J. West, *NOAA/NASA/DOD Geostorm Warning Mission*, JPL Internal Document D-13986, 18 October 1996.

[2] C. McInnes, *Solar Sailing: Technology, Dynamics, and Mission Applications*, 1999.

[3] J. West, *NMP ST-5 Solar Sail Project Concept Study Report*, July 15, 1999.

[4] C. McInnes, *Solar Sail Performance Requirements for the Geostorm Warning Mission*, University of Glasgow, DR 9623, 1996.

[5] G. Singh, *Attitude Control for the Geostorm Warning Mission*, JPL EM 3455-98-08, 12 October 1998.